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Genesis Mission Design

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Abstract

The Genesis spacecraft will collect solar wind samples from a halo orbit about the Sun-Earth L1 point for two years, returning those samples to Earth in 2003 for on-Earth analysis and examination. The solar wind will imbed itself into a set of ultra-pure material collectors that will be deployed throughout the collection phase of the mission. Analysis of the samples collected by the mission will contribute to our understanding of the origins of the solar system.

Introduction

The Genesis mission will be a "first of it's kind" mission. It will be the first U.S. mission to execute a robotic sample return and the first mission to return to the Earth from a halo orbit. It is the fifth mission selected as part of NASA's Discovery program and brings together the California Institute of Technology, the Jet Propulsion Laboratory, Lockheed-Martin Astronautics, Los Alamos National Laboratory, and Johnson Space Center.

The mission goal is to collect solar wind samples for a period of approximately two years and to return those samples to Earth for analysis (Rapp). The samples will be returned to the Utah Test and Training Range (UTTR) for mid-air helicopter recovery. Very little real-time science is planned for Genesis with the primary mission requirement being the sample collection.

The nominal launch date is January 7, 2001, with a return planned for August 19, 2003. The vehicle will be placed into the halo orbit on April 23, 2001 followed by 22 months of science collection .

The L1 libration point in the Sun-Earth system (between the sun and Earth) was selected as the platform for conducting the experiments since it provides uninterrupted access to the solar wind beyond the influence of the Earth's magnetosphere.

To collect the samples, a set of collector arrays will be deployed into the solar wind. The solar wind will imbed itself into these collectors and be stored for the return to Earth. Several different arrays will be available for collecting samples of different types of solar wind and will be deployed separately according to the type of wind that the spacecraft is experiencing. For low atomic number elements, such as oxygen, a sample concentrator will be used to collect sufficient material to exceed collector impurities.

Science Goals

The science investigations of the Genesis mission are based on the fact that the sun contains most of the mass of the solar system and, as such, its composition defines the average solar system composition. Differences between the composition of the sun and various other parts of the solar system (planets, comets, asteroids, meteorites, etc.) are very revealing as to the conditions that prevailed and the processes that occurred in the formation and evolution of the solar system.

Compilations of solar abundances are based mainly on analyses of meteorites and not on direct measurements of the sun. There are significant limitations to this approach, and accurate direct measurements of solar abundances are extremely important. Isotopic differences also exist among various planetary materials. Although these are much smaller than the elemental differences, they are of great importance because, in general, these would not occur with most of the chemical and physical processes that produce differences in elemental compositions. The isotopic compositions of solar matter define solar system averages and thus represent the starting point for the interpretations of the isotopic differences among planetary materials. Providing these data is the goal of the Genesis mission. The solar matter samples returned to Earth will lead to:

- A major improvement in our knowledge of the average chemical and isotopic composition of the solar system.
- A reservoir of solar material for 21st century science.
- Greatly improved models of the nebular processes by which planetary materials and the various bodies in the solar system (planets, comets, asteroids, Kuiper belt, unknown bodies, etc.) formed.

Mission Description

The Genesis mission design includes 22 months of solar wind collection outside the influence of the Earth's magnetosphere. This is accomplished via a mission timeline that spans just over 31 months, divided into five mission phases. A representative listing of the various mission phases is provided in Table 1. Figure 1 shows the spacecraft trajectory and representative mission events. These data are shown for the first launch date, January 7, 2001. The baseline 16 day launch period extends through January 22, 2001.

Table 1. Genesis Mission Phases

Phase	Time	Duration
Launch	01/07/01 – 02/06/01 L +0 – 30 d	30 d
Transfer	02/06/01 – 04/30/01 L +30 – 113 d	83
Science	04/30/01 – 03/22/03 L +113 – 804 d	691
Return	03/22/03 – 7/20/03 L +804 – 924 d	120
Recovery	07/20/03 - 8/20/03 L +924 – 955 d	31
where: L = Launch		

Launch Phase

The launch phase begins shortly prior to launch vehicle lift-off and ends 30 days after. The launch vehicle, a Delta 7326 with a Star37 upper stage, provides low energy injection (C3 = -0.6 km^2/s^2) into a transfer trajectory to a halo orbit about the Sun-Earth L1 libration point. Upon completion of the injection burn, the spacecraft will separate from the third stage of the launch vehicle, establish attitude control and communications, and perform initial subsystem checkouts. These activities must execute flawlessly to allow successful implementation of the first trajectory correction maneuver within 18 hours of launch, which is typical of halo orbit missions. Two more maneuver opportunities are planned within two weeks of launch to correct residual launch injection errors, if required.

To avoid contamination, the initial science activities are not performed until after the last significant launch correction maneuver. These activities include: opening the sample return capsule (SRC) to allow sufficient outgassing time prior to the start of solar wind collection; turning on solar wind monitors; and initiating the checkout of the flight software for autonomous monitoring of the solar wind regimes.

Additional spacecraft subsystem checkouts and calibrations are performed during the remainder of the launch phase. These include, but are not limited to, propulsion and attitude control calibrations considered essential for successful trajectory navigation.

Transfer Phase

The launch phase is followed by a three month period of fairly quiescent spacecraft activity. During this period, the sample return capsule continues to outgas and science algorithms continue to be exercised.

In order to prepare for the lissajous orbit insertion (LOI) maneuver (technically, the orbit is a lissajous orbit), the spacecraft activity level increases in the last few weeks of the transfer phase. The orbit insertion maneuver is the sole deterministic maneuver of the mission, ranging in size from 6 to 36 m/s, depending on the launch date. A halo orbit targeting maneuver is performed 10 days prior to LOI, if needed.

Following orbit insertion and the completion of outgassing, the science canister is opened, exposing all solar wind collection media. A brief period of end-to-end science instrument, collection media, and science algorithm checkouts precedes the end of the transfer phase and the start of the science phase.

Science Phase

As previously stated, solar wind acquisition is planned for just over 22 months while the spacecraft remains in orbit about L1. Solar wind

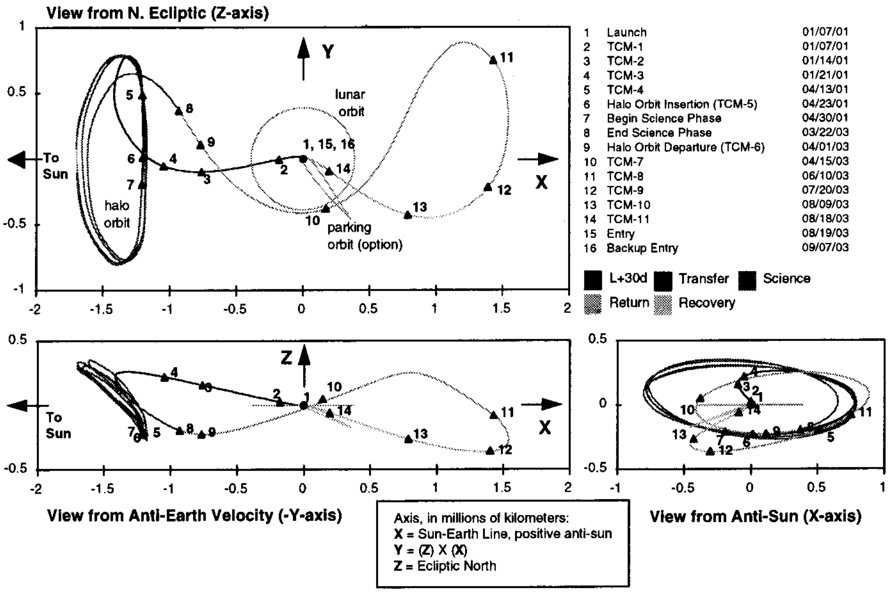

Figure 1. Spacecraft Trajectory

collection is accomplished via two, fixed, bulk collector arrays, a set of three, deployable, regime-specific collector arrays, and an electrostatic concentrator. Ion and electron monitors continuously measure solar wind conditions and provide data to on-board science algorithms that then control collector array and concentrator configuration. Solar wind data and instrument configuration are stored on-board and transmitted to Earth via weekly downlinks.

During the science phase, the spacecraft is oriented such that its spin axis, which is normal to the collectors, points toward the prevailing direction of the solar wind, approximately 4.5 degrees forward of the sun. Daily precession maneuvers, or turns (about 1 degree in size), are performed to track the sun and maintain the desired orientation.

The spacecraft remains in orbit about L1 for four revolutions. Eight to sixteen statistical station-keeping maneuvers are anticipated during this time.

Return Phase

After remaining in orbit about L1 for four revolutions, the spacecraft flies through a heteroclinic dynamical channel connecting the L1 and L2 regions and loops around L2 to set up for a daylight return to Earth. Three statistical maneuvers are planned for this phase.

At the beginning of the phase, all science instrumentation is turned off, and the science canister and return capsule lids are closed. It is expected that the storage canister and the sample return capsule will each be opened and closed only once during the mission. The spacecraft configuration remains unchanged for the remainder of the mission. Propulsion and attitude control subsystem calibrations that are critical for terminal navigation are repeated prior to the first return maneuver.

Recovery Phase

The recovery phase starts 30 days prior to Earth return. Three statistical maneuvers are planned for this phase. The last two are the primary entry targeting maneuvers and are performed at entry minus 10 days and minus 1 day.

About four hours before Earth entry, the spacecraft reorients to the sample return capsule release attitude, spins up to 15 rpm and releases the capsule. Soon after release, the spacecraft reorients to point its thrusters to Earth and performs a maneuver which will cause the spacecraft to enter Earth's atmosphere, but break- up over the Pacific Ocean.

Following release from the spacecraft, the Genesis sample return capsule experiences a passive, spin-stabilized aero-ballistic entry, similar to that of the Stardust mission. When the capsule has decelerated to 1.4 times the speed of sound, the on-board avionics system fires a

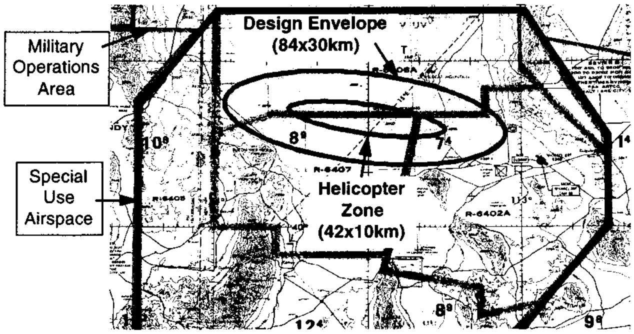

Figure 2. UTTR footprints

mortar to deploy the drogue parachute. The drogue is a disk-gap-band design, with heritage dating to the Viking program, and an extensive history of supersonic applications. It serves both to increase the deceleration of the capsule, and to stabilize it through the transonic phase. As the capsule descends into the airspace of the Utah Test & Training Range (UTTR), recovery helicopters are directed to fly toward the intercept point. The capsule's ballistic path is designed for delivery within an 84 x 30 km footprint with subsequent reduction to a 42 x 10 km helicopter zone, see Figure 2. The first helicopter on-site will line up and match descent rate, then execute a Mid-Air Retrieval (MAR) capture. If a pass is aborted for any reason the pilot can line up and repeat.

Mid-air retrieval uses an aircraft to intercept and capture a payload descending under a parachute, so that it can be returned to a designated location without impact damage. The basic technology dates to the 1920's, when Pennsylvania dentist Lytle S. Adams developed a system for aircraft pickup of parcels from the ground, without landing. By 1938, All American Aviation, founded partly by Dr. Adams, was using his method to pickup mailbags along dangerous mountain routes in Pennsylvania, Ohio, and West Virginia. Similar rigs were used in World War II to rescue downed fliers from difficult terrain. In 1955, a spin-off company, All American Engineering, adapted the principles to mid-air capture of parachute loads. The US Air Force first used MAR in August 1960, to capture the instrument payloads of Discoverer satellites over the Pacific Missile Test Range. In the 60's and 70's, film canisters from reconnaissance satellites were routinely recovered by both fixed wing aircraft and helicopters.

The MAR helicopter subsystem, see Figure 3, consists of a constant tension winch, a catch pole, and a hook and release mechanism. The engagement is performed by flying a low pass, dragging the pole through the canopy. The fabric of the canopy folds around the pole momentarily, then slides down the pole and into the mouth of the hook. As the hook takes the load, it is pulled free of its attachment to the pole, transferring the load to the retrieval cable. On contact, the winch allows cable to spool out, limiting the engagement load.

Once the capture is made, the recovery helicopter will fly to the landing site, Michael

Figure 3. Mid-Air Retrieval By Helicopter

Army Air Field, with the SRC in tow. The SRC will be lowered directly into a handling fixture for post-recovery processing and transfer to the NASA Curatorial Facility at Johnson Space Center.

Spacecraft Design

The Genesis spacecraft uses considerable hardware and software inherited from previous spacecraft projects, such as Stardust and Mars Surveyor 98, orbiter and lander. These lessons learned, from design through operations, provide low cost and low risk, which is necessary in the "faster, better, cheaper" paradigm.

The spacecraft spins at 1.6 rpm, a balance that allows adequate spin stabilization, yet is consistent with performance of the science sensors and on-board navigation. Figure 4 illustrates the spacecraft forward deck sun pointing configuration. Two solar arrays are deployed after launch and separation from the Delta 7326 launch vehicle. This configuration is for nominal science operations, with the science canister and SRC backshell opened. S/C mass is about 643 kg, including 143 kg of hydrazine fuel contained in two tanks for the blowdown propulsion system. This provides about 450 m/s ΔV capability.

Figure 5 illustrates the aft deck, which points toward Earth for the majority of the mission. There are two redundant strings of thrusters on the deck which include a total of four 22 N thrusters for large translational maneuvers, and eight 1 N thrusters for small maneuvers, adjusting the spin rate, and spin axis orientation (precession). All thrusters are on the aft deck to avoid contamination of the forward deck science instruments. Nutation caused by perturbing torques, such as uncoupled thrusters, is passively damped by two small rings filled with viscous fluid.

Attitude knowledge is provided by one of two star cameras. A self contained star catalog is used with a star pattern recognition algorithm to provide an inertial quaternion as output to the flight processor. Digital two axis sun sensors facing forward, and spinning sun sensors perpendicular to the spin axis, provide backup spin rate and axis knowledge.

For S-band telecommunications, the spacecraft has two patch low gain antenna (LGA), facing forward and aft, to provide low data rate Earth communications and tracking. A single helix medium gain antenna (MGA) on the aft deck is used during the science phase for high data rate communications.

Solar arrays, three square meters in size, provide 265 watts of regulated power. When the spacecraft is not pointing to the sun, such as for trajectory correction maneuvers, a single 16 amp-hr battery provides power, limiting the off-sun time to about 80 minutes.

The Command and Data Handling subsystem has redundant 10 Mhz processors, and more than 95 Mbytes allocated for downlink science and engineering data storage. Fault protection software (FPS) continually monitors spacecraft health and status. Fault protection is responsible for switching from primary to redundant strings, and, if needed, triggering entry into an autonomous safe mode which reconfigures the spacecraft to minimize electrical power loads, and then precesses quickly to sun pointing.

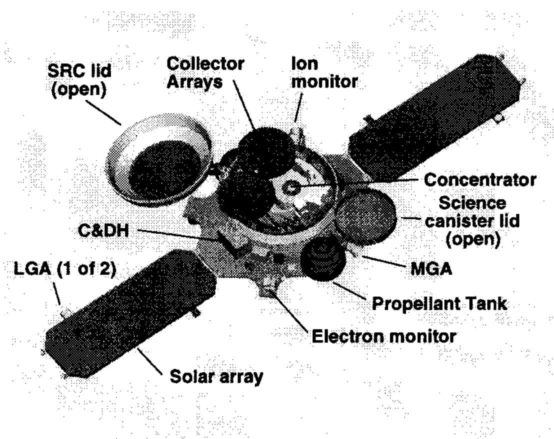

Figure 4. Forward Deck View Pointing Toward Sun

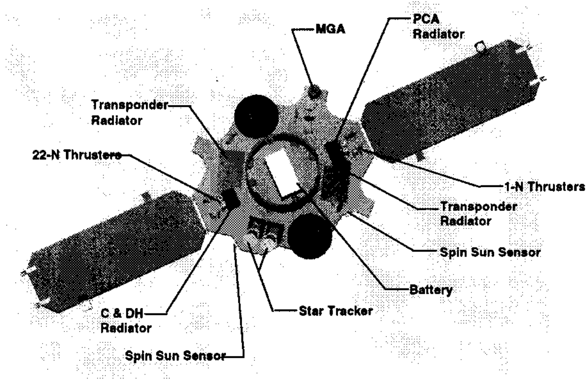

Figure 5. Rear Deck View Facing Earth

Payload

The science payload for the mission includes three primary components: the solar wind monitors, the collector arrays, and a concentrator.

Collector Arrays
The vehicle will carry five separate solar wind collectors, each approximately 73 cm in diameter. Two of the collectors, designated bulk arrays, will be exposed to the solar wind during the entire science collection phase of the mission. One of the bulk collector arrays is located inside of the canister cover. The second bulk array is the top cover of the regime collector array stack. The regime-specific collectors will be exposed to the solar wind individually according to the type of solar wind that is impinging on the vehicle at the time. Each collector is made up of 55 individual, ultra-pure material, hexagonal wafers. Most of the wafers will be made of silicon, approximately 0.5 mm thick. However, other materials such as high purity aluminum and gold foils will also be used since different types of solar wind are best collected with different types of materials. The collector array stack is illustrated in Figure 6.

Solar Wind Monitors
In order to collect different types of solar wind on different collector arrays, the spacecraft carries two solar wind monitors: an ion monitor and an electron monitor. These monitors are used to detect the type of solar wind that is impinging on the vehicle. When the type of solar wind changes, the exposed collector will be returned to the stack in favor of the proper collector for the type of solar wind. It is expected that the type of solar wind will change approximately every 3-4 days, necessitating switching the exposed collector. Collector switching will be performed by on-board software. The solar wind monitor locations are illustrated in Figure 4.

Concentrator
Since the total mass of solar wind that will be collected is very small relative to the size of the collector arrays, a solar wind concentrator, Figure 7, will also be used in an attempt to collect a concentrated sample of the solar wind. In this way, a higher signal-to-noise ratio between the solar wind and any contamination

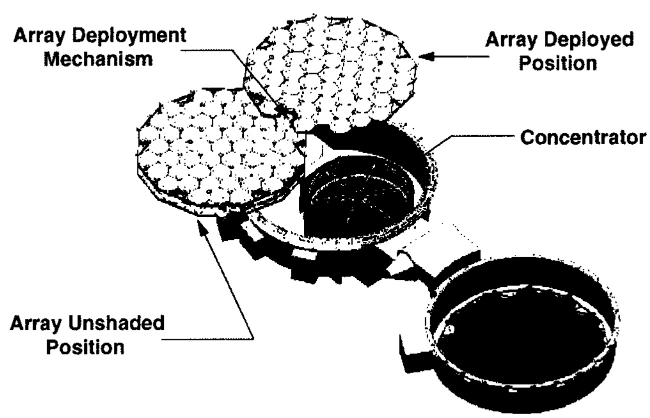

Figure 6. Collector Arrays

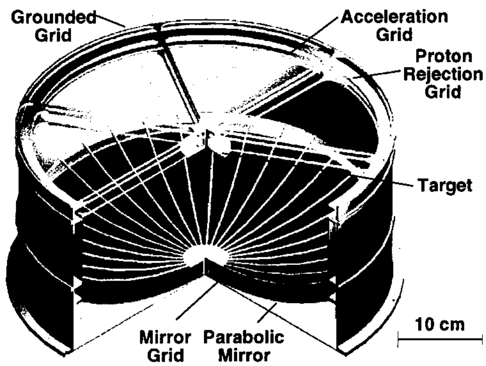

Figure 7. Concentrator

from the collection process is obtained and collector material impurities will have less influence on the analysis of the solar wind particles. The concentrator will be exposed to the solar wind during the entire collection phase of the mission. It is expected that the array that is part of the concentrator (separate from the other five individual arrays) will collect approximately 20 times more solar wind particles than the unconcentrated arrays. This is particularly important in the case of oxygen (and other elements of atomic number 6 through 22), for which the concentrator is expected to provide the required fluence enhancement.

Sample Return Capsule

The sample return capsule is the key component of the flight system that allows successful return of the solar wind samples without having to retrieve the entire spacecraft. The five collector arrays and the concentrator are stored inside the science canister which in turn is inside the return capsule. The return capsule protects the science canister from the extreme aerodynamic heating of atmospheric entry. The capsule is illustrated in Figure 8.

The return capsule is designed with a 60 degree half-angle blunt cone constructed from phenolic impregnated carbon ablator. The backshell is also protected from the heat of atmospheric entry as it is covered with a super lightweight ablator material. The capsule shape provides a 70 kg/m^2 ballistic entry coefficient. The capsule's current mass is 193 kg.

Avionics and electronics are powered by two primary cell batteries. The avionics are responsible for controlling all capsule entry and descent activities including parachute deployment and descent tracking electronics. Accelerometers are used to deploy a drogue chute. The main chute is deployed via a timer initiated at drogue chute deployment with a barometric pressure switch backup.

Three locating and tracking methods are implemented on board the sample return capsule. The primary method relies on skin tracking of

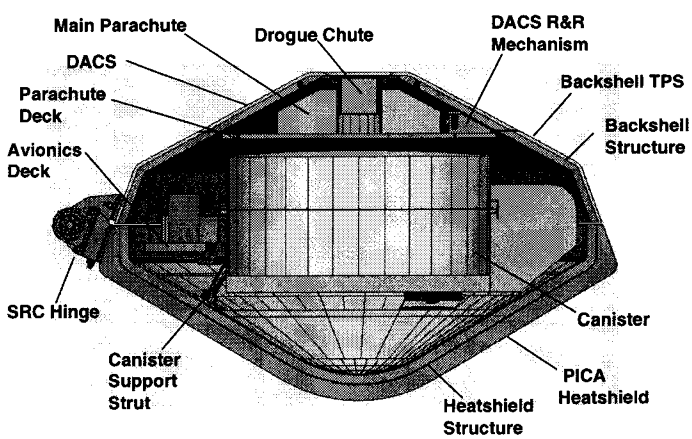

Figure 8. Sample Return Capsule

the capsule through local radars at the Utah Test and Training Range. This is backed up by a digital communications network system that relies on the Global Positioning Satellites, an on-board L-band antenna, and controller to resolve latitude, longitude, altitude and direction. The digital network also contains a UHF transmitter with a corresponding ground receiver. The last method for locating the return capsule is a VHF beacon which can operate for a maximum of 40 hours. This method is most useful in the event of a ground impact.

Trajectory Design

The nominal trajectory includes all of the standard pieces that would be expected in any sample return mission, a launch segment, a collection orbit, and a return leg; however, the techniques used to design the pieces represent an innovative approach to trajectory design. The Genesis trajectory is the first mission to be designed using modern dynamical systems theory. During its early design phase, the SOHO mission spearheaded the application of dynamical systems theory to halo orbit design (Gomez), but the actual mission is using the classic methods developed for the ISEE3 mission (Farquhar 1977, 1978, 1980; Jordan).

While the design and construction of the libration point orbit itself is well understood, the computation of low energy return trajectories is a more challenging design problem. The near-optimal Genesis trajectory was found by computing and understanding the characteristics of the invariant manifolds associated with the halo orbit (Howell, et al). The transfer trajectory was constructed using the stable manifold (Figure 9); the free return trajectory was constructed using the unstable manifold (Figure 10).

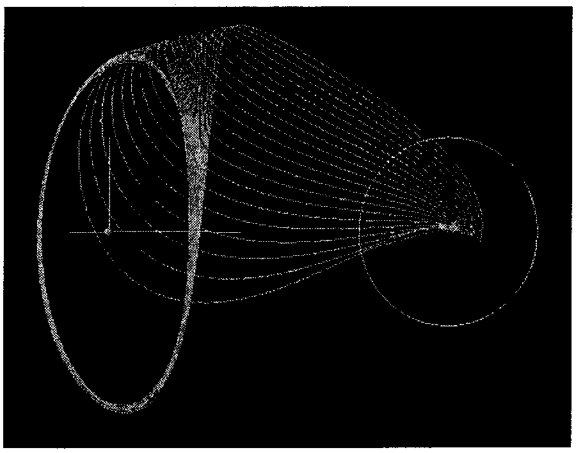

Figure 9. Stable Manifold

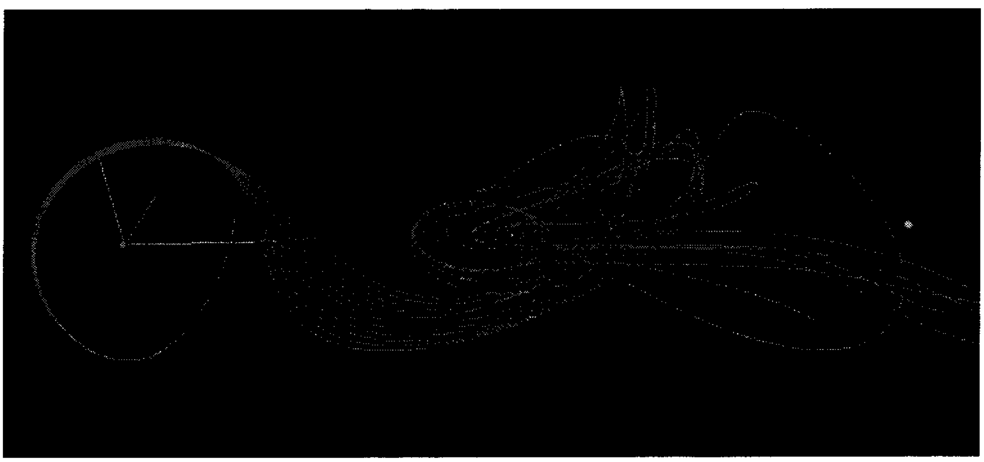

Figure 10. Unstable Manifold

One of the first unusual features of the mission is that there is only one deterministic ΔV of 6 m/s. The trajectory begins with a standard launch phase followed by the deterministic injection ΔV that inserts the vehicle into the halo orbit. However, following the completion of the halo orbit phase of the mission, the vehicle automatically leaves the libration point region with no departure maneuver. In addition, following that departure, the vehicles goes past the Earth into the region near the L2 libration point (on the far side of the Earth) before returning to the Earth with the final portion of the return leg. This is necessary in order to satisfy the requirement for a day-side return to UTTR. This free transfer exploits the homoclinic behavior of the L1 and L2 regions (see Barden). This means that there are gravitational channels, like the jet streams of our atmosphere, providing transport between the regions interior and exterior to the Earth's orbit with L1 and L2 as gate keepers. This transport mechanism is what governs the temporary capture of Jupiter comets (Lo) and provides the free return of the Genesis mission.

Launch & Transfer Phases

The three sigma launch error from the launch vehicle is estimated to be 21 m/s. However, this error is expected to go down as more data becomes available for this launch vehicle configuration. As a result of launch dispersions and the sensitivity of the transfer trajectory, a trajectory correction maneuver will be executed within 18 hours after launch to cleanup any errors from the launch. This is typical of halo orbit missions with direct launch into halo orbits.

The launch period for the mission is extremely flexible, allowing launch opportunities for approximately six months per year. Launch dates that provide a ΔV budget within the spacecraft capability are available for 13 days in December 2000, 17 days in Jan 2001, and 17 days in February 2001. Although many different launch dates are available, all launch trajectories for any single month connect to the same halo orbit and return trajectory. Changes in the LOI maneuver are used to accommodate different transfer trajectories. The transfer trajectories were selected as trajectories along the stable manifold that is associated with the halo orbit.

Science Phase

About three months after launch, the spacecraft is inserted into the halo orbit with the single deterministic maneuver. The halo orbit is a northern halo orbit (class I) with a Y-amplitude of nearly 800,000 km. The excursion of the orbit out of the ecliptic plane (the Z-amplitude) is approximately 300,000 km. Two to four station keeping maneuvers per halo orbit revolution are required to maintain the orbit.

Return Phase

As previously noted, the vehicle leaves the libration point region with no departure maneuver to begin the return to the Earth. Since the return was designed as a segment of the unstable manifold associated with the halo orbit, the orbit naturally unwinds from the halo shape. The requirement to return to UTTR for a mid-air helicopter recovery requires that the vehicle return during daylight hours. A trajectory from the unstable manifold that provides a dayside Earth return was found; however, it requires that the vehicle first enter the L2 region (on the far side of the Earth) before returning to the Earth. The region of space that can be reached with the unstable manifolds of the orbit is vast; however, the segment of the manifold that provides an acceptable return to the Earth is extremely limited. The Earth return will require very precise targeting of a very narrow and very unstable corridor of space.

A backup entry opportunity has been designed for the mission, in the event that the primary reentry cannot be executed. This is achieved by capturing the spacecraft into a highly elliptic parking orbit (19 day period) with a single backup reentry opportunity planned for September 7, 2003.

ΔV Budget

The Genesis ΔV budget estimate provided in Table 2 is very conservative. The ΔV margin at PDR (6/20/98) is shown below (18%). The conservatism is allocated to the launch error correction, the return station keeping, and the ACS propellant budget. As the design and analysis of the various subsystems are refined and more accurate estimates can be provided, the ΔV margin is expected to increase to about 30% at launch.

Navigation

The navigation process for Genesis includes trajectory estimation, prediction and trajectory control through propulsive maneuvers. The trajectory estimation will be performed using JPL's deep space orbit determination and precision trajectory computation software

Table 2. Current Genesis ΔV Budget

Mission Events	**ΔV (m/s)**
Launch Error Correction (95%)	93
LOI	6 to 36
Halo Station Keeping	24
Return Station Keeping	45
Primary Entry Target	4
Deboost S/C	20
ACS	71
Backup Entry	87
Margin	70
Total	**450**

system, DPTRAJ/ODP, which has been used extensively on all previous NASA deep space exploration missions. NASA's Deep Space Tracking Network (DSN) will provide S-band Doppler and range measurements to the navigation filter for estimating the trajectory. The navigation operation will be patterned closely after that used for NASA's Stardust Discovery mission, to gain economy of operations through multi-mission use of facilities and personnel. Since the Genesis mission has a spinning spacecraft design, there will necessarily be some differences between the two operations, especially in maneuver planning and implementation.

There are specific Genesis mission characteristics that drive the navigation design. These include both the baseline spacecraft design and the mission profile. For instance, the maneuver calculations for the spinning spacecraft must take into account the unbalanced thruster design which was chosen to minimize contamination of the science collectors and sensors. This implies that precession turns and spin changes will result in a change in translational velocity. These changes will be accommodated in trajectory correction maneuvers so that the vector sum of all the planned turn-spin-burn activities will give the desired resultant delta-velocity. From the mission profile perspective, the navigation function must maintain the desired trajectory by correcting the injection errors, delivering the spacecraft to and maintaining it in the halo orbit, returning to Earth, and guiding the spacecraft to the release point so that the entry vehicle arrives within tolerances at the recovery site in Utah.

Preliminary error analysis focused on the effect of non-gravitational acceleration errors and maneuver errors caused by daily precession maneuvers during the approximate two year science gathering period in the Lissajous orbit. Only radio-metric tracking data were used for the orbit determination of the mission. Assumed data types were S-band Doppler and Ranging. Data noise values of 1 mm/sec for the Doppler data and 200 m for the range were assumed. The proposed DSN tracking was assumed to be rather sparse, tracking only ten hours per week from two stations every other day. Non-gravitational stochastic acceleration uncertainties in steady state were assumed with a one sigma value of $5x10^{-12}$ km/s^2. For this phase, data arcs over 60 days long were required before the spacecraft position and velocity estimates reached the steady state values. This behavior is mainly due to the sparse tracking schedule and the perturbations due to the daily precession maneuvers which had an assumed random error of 1 mm/s each. This level of precision will require that the spacecraft maneuvering system be calibrated in flight by using DSN tracking. The study concluded the one sigma value of position uncertainty is 4 ~ 8 km in steady state, and the corresponding velocity uncertainties are less than 5 mm/s.

The effective navigation errors for maneuver planning are shown in Tables 3 and 4. These uncertainties are given in the A,V,N coordinate frame, where the V unit vector points along the instantaneous velocity vector of the spacecraft, the axis N is normal to the plane containing the Earth-relative position and the velocity vectors of the spacecraft, and the axial vector A completes the triad, with A = V x N. Note that during the Lissajous orbit, the A direction is almost along the line-of-sight from the Earth. As seen in the table, the orbit determination error grows substantially as time elapses between the last data point in the estimate (the data cutoff) and the time a planned event (i.e., a propulsive

Table 3. Orbit Position Uncertainties in Science Phase (1 sigma value)

Time	**Position Error (km)**		
From ΔV	σ_A	σ_V	σ_N
ΔV-1 wk	2.64	1.88	7.10
ΔV-2 wk	6.73	3.34	13.4
ΔV-4 wk	25.1	15.2	29.6

Table 4. Orbit Velocity Uncertainties in Science Phase (1 sigma value)

Time	**Velocity Error (mm/s)**		
From ΔV	σ_{dA}	σ_{dV}	σ_{dN}
ΔV-1 wk	4.23	3.22	5.55
ΔV-2 wk	7.31	5.29	5.91
ΔV-4 wk	22.0	17.8	6.23

maneuver or ΔV) occurs. These results indicate maneuver planning and execution should be performed within two weeks of the last data used to determine the trajectory.

The other critical phase for navigation is the Earth entry. Here the planned scenario is to increase the nominal DSN tracking interval to eight hours of Doppler and range measurements per day at twenty days prior to Earth entry. This continues up to three days before entry at which time continuous tracking begins. This tracking supports orbit determination deliveries to the entry targeting maneuvers planned at ten days and one day prior to entry. The critical parameter on entry which determines the entry accuracy is the flight path angle, the angle made by the tangent to the trajectory and the local horizon at 125 km altitude. Orbit determination simulations for this interval indicate the uncertainty in the flight path angle is less than 0.02° (three sigma). When root-sum-squared with the maneuver execution contributions, the total flight path angle error for entry resulting from the last maneuver at entry minus one day is 0.05° (three sigma). This is the same as the current requirement for navigation accuracy on entry, so further navigation development as the project matures to flight operations must avoid growth in the uncertainty of this parameter.

In summary, the orbit determination performance during the science collection phase with the daily precession maneuvers is adequate given the assumptions of maneuver calibrations and maneuver predictability. Maneuver perturbation analyses indicate that the Lissajous orbit and its return to Earth are not highly sensitive and small orbit corrections can correct the expected spacecraft state errors. In addition, preliminary maneuver analysis has been performed to measure the impact of unbalanced thrusters on the Lissajous orbit control and although these are on-going, initial results show there are engineering solutions to reduce their effect, such as turning a complete 360° (or in some cases 720°) during the precession parts of the burn to cancel the precession translational velocity changes. These solutions use extra fuel but reduce the overall uncertainty in the delivered delta-velocity.

Mission Operations

Genesis will employ the concept of distributed operations to achieve its mission objectives. Distributed operations places responsibility of tasks and decisions in the hands of the experts, who remain at their home institutions. The project organization is constructed from five different institutions, each providing unique and special skills.

In tune with Discovery mission guidelines, the Principal Investigator is responsible to NASA for all aspects of the mission. For Genesis, this individual resides at The California Institute of Technology and also leads the Science Team, which is comprised of over 20 co-investigators.

Lockheed Martin Astronautics provides expertise for the design and construction of the spacecraft bus and sample return capsule. Spacecraft and payload integration will also be conducted at LMA. During flight, LMA will be responsible for the health and safety of all spacecraft systems.

Los Alamos National Laboratory has been selected to develop part of the science payload: the electron monitor, the ion monitor and the concentrator. Experience with many instruments involved with solar experiments make them the ideal partners for this task. LANL also is planned to support solar wind monitoring during mission operations.

The Jet Propulsion Laboratory contributes overall day-to-day management of the project. JPL is also responsible for development of the sample collector arrays, science canister, and integration of the science payload. Finally, JPL will provide the mission design, mission planning, navigation and mission operations functions.

Johnson Space Center is known for its extensive and long recognized expertise in contamination control and sample curation. Given the nature of the Genesis mission, JSC is selected to fulfill this task.

Summary

Built on the partnership between the California Institute of Technology, the Jet Propulsion Laboratory, Lockheed-Martin Astronautics, Los Alamos National Laboratory, and Johnson Space Center, the Genesis mission will be the first U.S. mission to execute a robotic sample return and the first mission to return to Earth from a halo orbit.

The spacecraft design and mission operations rely heavily on systems developed from previous projects. This inheritance, however, provides lessons learned, from design through operations, that keep the mission in the low cost, low risk genre required in the faster, better, cheaper paradigm of the Discovery program. The mission design offers unique challenges that are critical to the mission's success. To ensure successful return of the flight system to Earth, the Genesis mission design will require the development of new techniques and the application of old techniques in new ways to new problems. The challenges are found in the trajectory design and navigation in this sensitive region of space to return the samples back to Earth.

The Genesis mission is an exciting venture that brings together expertise from academia, industry and government to develop a low cost mission with an important science objective under NASA's directive for better, faster, and cheaper missions. Genesis will provide the first raw materials from the sun for 21st century science to increase our knowledge of the sun and address the questions of the origins of our Solar System.

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